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INVERSE TRANSONIC AIRFOIL DESIGN METHODS INCLUDING BOUNDARY LAYER AND VISCOUS INTERACTION EFFECTS

Semiannual Progress Report

September 1, 1983 -- January 31, 1984

College of Engineering
Texas A&M University

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INVERSE TRANSONIC AIRFOIL DESIGN METHODS INCLUDING BOUNDARY LAYER AND VISCOUS INTERACTION EFFECTS

I. Introduction

This report covers the period 1 September 1983 to 31 January 1984. The primary task during this reporting period was the continued development of the massive separation model and computer code (SKANSEP). In particular, detailed investigations were conducted with the boundary layer displacement surface correction technique discovered near the end of the last reporting period. This report will present detailed results using this technique and show comparisons with experimental data.

II. Personnel

The staff assigned to this project during the present reporting period were:

Leland A. Carlson, Principal Investigator
September--January--Approximately 1/8 time

Unfortunately, due to lack of payroll money remaining in the project account, no undergraduate or graduate student was available during this period to work on the project.

III. Massive Separation Studies

During the past six months, this research effort has concentrated on investigating the performance of the massive separation direct-inverse computer

program SKANSEP at subcritical conditions. This program is designed to handle massive separated flow at high lift conditions and uses a simplified Kuhn-Nielsen boundary layer method for the turbulent portion and a modified compressible Thwaites scheme for the laminar part.

Near the end of the last reporting period, several significant changes were introduced into SKANSEP. These changes primarily concerned the calculation of the displacement thickness on the upper surface in the last fifteen percent of chord and the manner in which the location of the separation point was determined. Unfortunately, time only permitted the inclusion in the last progress report[1] of one test case. While this case did show reasonable agreement with experiment, it did not establish the general validity of the method and further testing was needed. During this reporting period, extensive tests of the corrected program have been conducted and the results compared to other theoretical results and to experimental data from the NASA Langley Low Turbulence Pressure Tunnel. In the remainder of this section, these results will be presented and discussed in detail.

Figure 1 presents results obtained with SKANSEP for each of its two modes of operation. These modes are selected by the user via the input data and determine the method in which the pressure is computed in the separated region. In the KSEP=0 case, the pressure is assumed to be constant in the separated zone; and this option should be applicable to incompressible cases. At higher freestream Mach numbers experimental data indicates that the surface pressure in the separated zone actually increases in a monotonic nonlinear fashion from the separation point to the trailing edge, and this possibility has been included in SKANSEP via the KSEP=1 option. The actual mechanism for calculating the separated pressure variation in this case follows the analytical derivation of Barnwell presented in Reference 2.

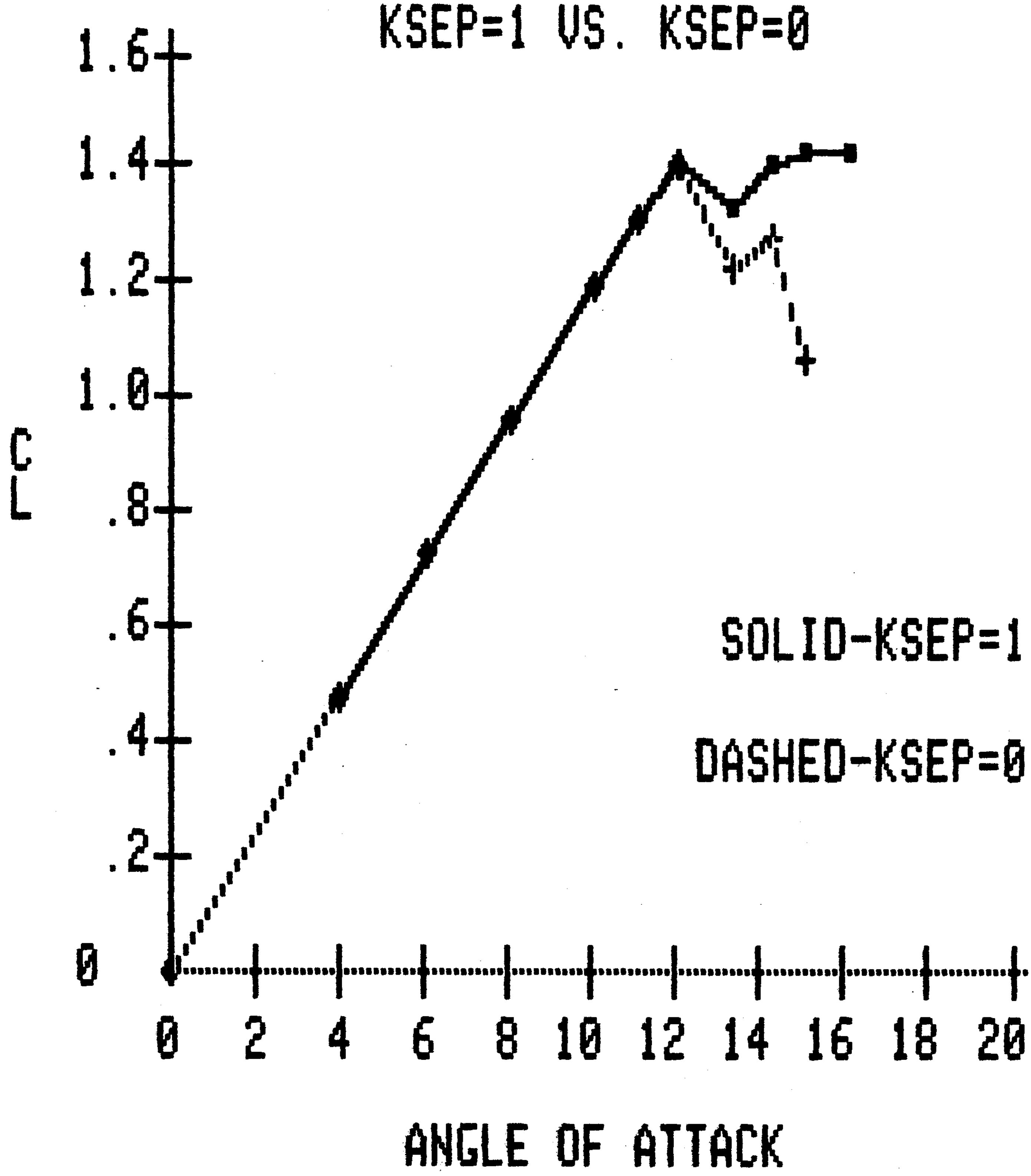

Figure 1 -- Lift Curves for the Two Separated Pressure Options
(NACA 0012, Mach No. = 0.3, Reynolds No. = 6 million)

The results shown on Figure 1, which are for a NACA 0012 airfoil, a nominal freestream Mach number of 0.3, and a Reynolds number based on chord of six million, indicate that at high angles-of-attack there are significant differences between the the two options. Both options indicate significant separation effects at angles-of-attack greater than 12 degrees, but the KSEP=0 (constant separated pressure) predicts a sharper stall break and decrease in lift coefficient. Interestingly, both options agree up to the point of significant separation. This latter agreement should exist, but is was not actually achieved in earlier versions of SKANSEP. Thus, the present corrected version of the computer model and code is a significant improvement over earlier versions of the program. It should be noted that both models predict a maximum lift coefficient of about 1.40. Further, for this compressible case, examination of experimental data indicates that the variable separation pressure model (KSEP=1) should be the better of the two options.

In the present version of SKANSEP, the theoretical lift coefficient for a given case is actually computed twice--once from the circulation at the trailing edge and again from integration of the computed pressures on the airfoil surface. Since most good experimental data for these conditions obtained lift coefficient from surface pressure integration, it is essential that the model accurately obtain lift from the computed pressures in order to permit valid comparisons. In the past, particularly at medium and high angles-of-attack, SKANSEP had yielded significantly different values for the CL obtained from circulation from that computed from surface pressures.

Consequently, the method of computing lift from surface pressure in SKANSEP was carefully investigated. The normal procedure in this case is to compute via integration a normal force coefficient and an axial force coefficient, and then utilizing the angle-of-attack resolve these forces into lift and pressure drag

coefficients. Since the axial force computation involves the vertical airfoil ordinates, it is normally the more inaccurate of the two values; and at high angle-of-attack this inaccuracy in axial force can significantly affect the accuracy of the resultant lift coefficient. However, for the subcritical cases considered here (i.e. Mach 0.3 and Reynolds number of 6 million) the pressure drag should be zero. Thus, each case was examined and the axial force coefficient corrected so as to yield a zero pressure drag. The resultant corrected axial force coefficient was then used in conjunction with the normal force coefficient to compute a corrected, and hopefully more accurate, value for the lift coefficient from pressure integration.

The results of this correction are shown on Figure 2 and compared to the lift coefficients predicted by the circulation. On the figure, CLCIR refers to the lift via circulation while CLP denotes the corrected lift coefficient determined by pressure integration. As can be seen the two values are in excellent agreement up to about 14 degrees angle-of-attack, which is a significant improvement over previous calculations. However, at the very highest angles-of-attack, the two values are different. In those cases, the circulation lift coefficient tends to remain near its maximum values, while the pressure lift coefficient shows a definite loss of lift. Quite possibly, the occurrence of these differences is an indicator that the flow is "stalled" and unsteady.

In assessing the validity of any method, it is always beneficial to make comparisons with results obtained by other theoretical methods. Figure 3 compares CLP results obtained with the present method with values recently presented by Anderson, Thomas, and Rumsey[3] using the thin layer Navier-Stokes equations. While the latter were computed at a Reynolds number of one million rather than the present condition of six million, they should be a valuable

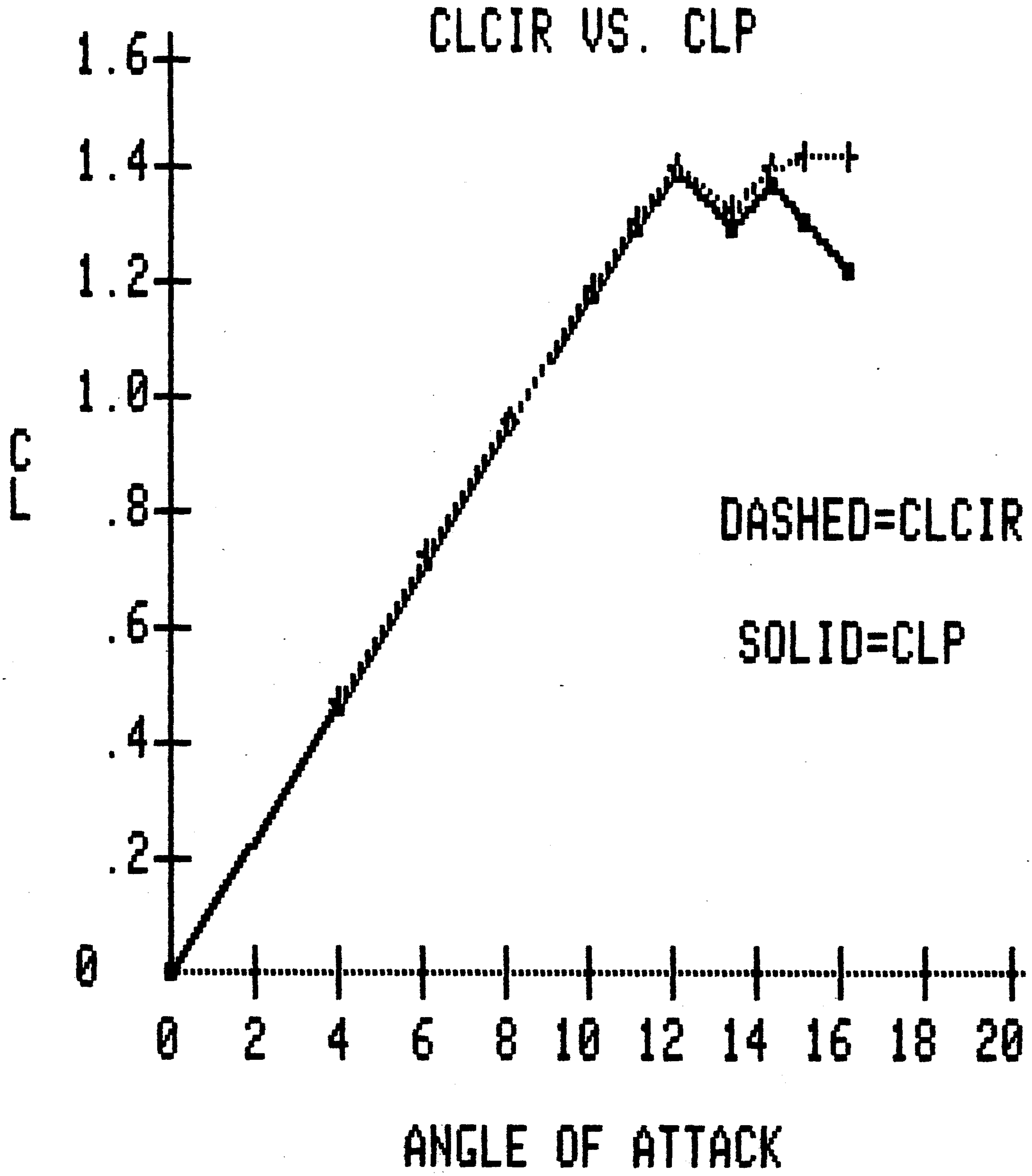

Figure 2 -- Comparison of Lifts Obtained from Circulation and Pressure Integration (NACA 0012, Mach No. = 0.3, Reynolds No. = 6 million)

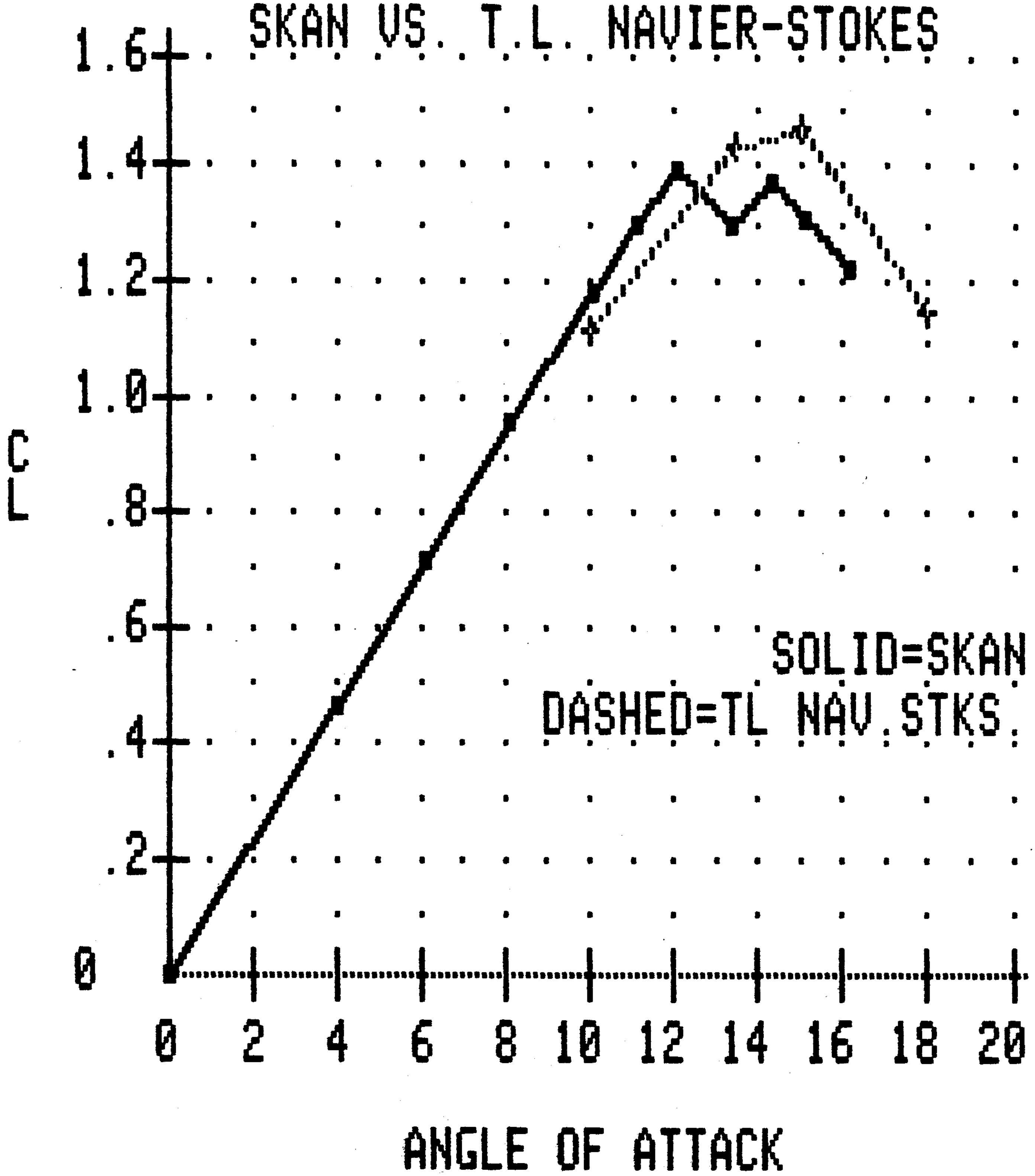

Figure 3 -- Comparison of SKAN and Thin-Layer Navier Stokes Results (NACA 0012, Mach No. = 0.3)

comparison. As can be seen on the figure, the two sets of data are in reasonable agreement. However, the thin layer Navier-Stokes results do predict a higher maximum value for lift coefficient and indicate a higher angle-of-attack for the stall break. This behavior is somewhat surprising, since most experimental data indicates a lower maximum lift coefficient and an earlier stall break at lower Reynolds numbers. However, if CLCIR values from the present results had been plotted instead of CLP, the agreement would have been even better. Finally, it should be noted that the thin layer Navier-Stokes CL value plotted at 18 degrees is an average value. As pointed out in Reference 3, the solution of this case never converged to a steady value but oscillated in a stable manner, indicating possibly that the flowfield behavior after the stall break is highly unsteady.

Based on experimental data for the NACA 0012, separation on the upper surface should initially start at the trailing edge. Then, as angle-of-attack is increased, the separation point should move steadily forward on the airfoil. Figure 4 shows that the present theoretical method does indeed predict this type of behavior. As shown on the figure, the theory indicates that there is no upper surface separation at angles-of-attack less than or equal to twelve degrees. Between twelve and fourteen degrees the separation point moves slowly forward, and at higher angles-of-attack it moves rapidly forward. In fact, at sixteen degrees, approximately seventy percent of the upper surface is separated according to the present theory. In examining these results, it should be noted that frequently the location of the separation point is determined on the medium grid; and thus the accuracy of this location is probably only within three percent. This slight uncertainty in the separation point location can induce or create some oscillations in the theoretical lift curve which probably would not be present in corresponding experimental data.

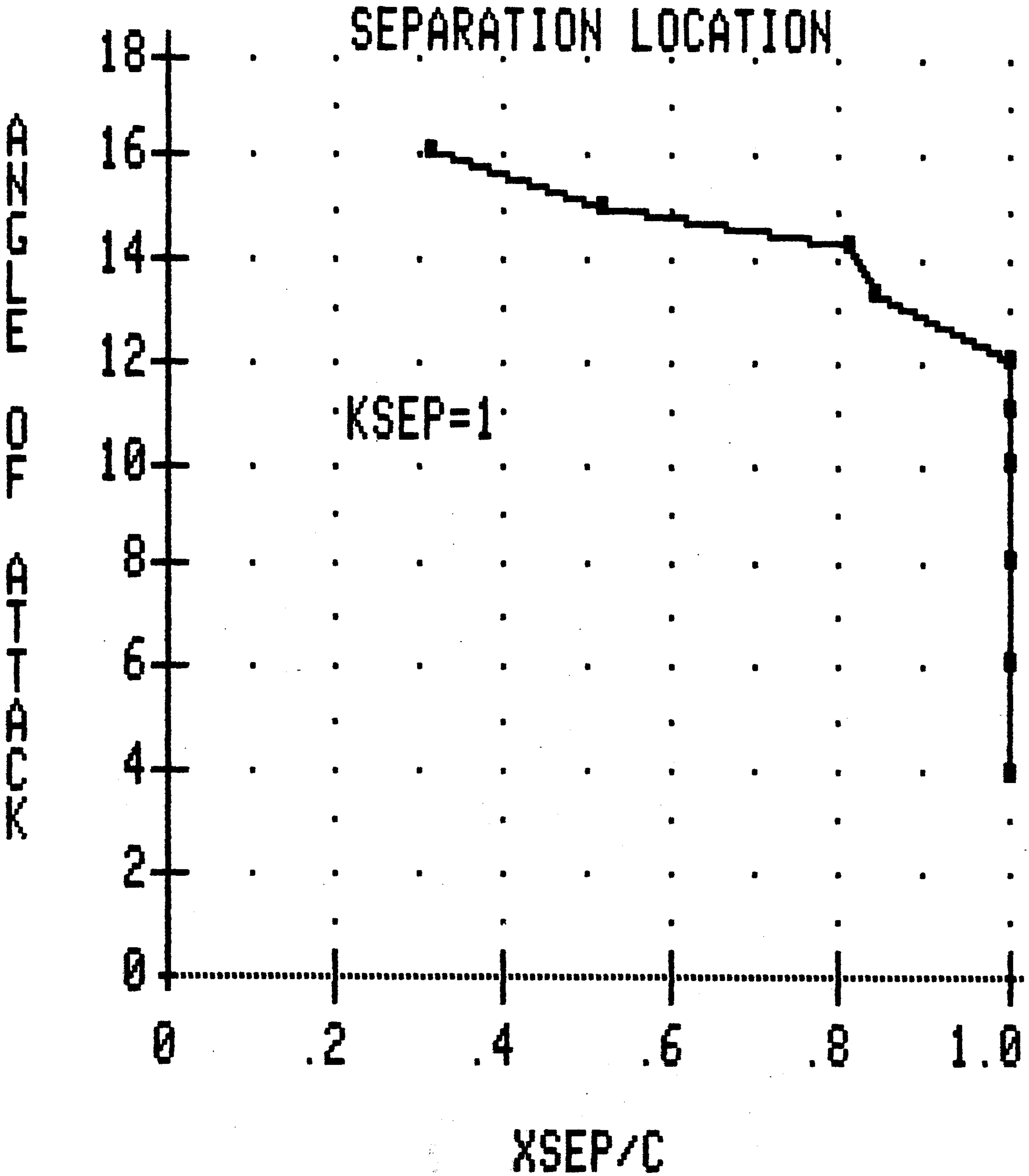

Figure 4 -- Variation of Upper Surface Separation Point with Angle-of-Attack
(NACA 0012, Mach No. = 0.3, Reynolds No. = 6 million)

Figure 5 compares results obtained with the present version of SKANSEP for a NACA 0012 at a nominal Mach number 0.3 and Reynolds number of 6 million with data obtained in the NASA Langley Low Turbulence Pressure Tunnel. For this figure, the theoretical lifts are those due to circulation; and the experimental data is that obtained with no boundary layer trips on the airfoil. Here the experimental data indicates a miximum lift coefficient of about 1.40, followed by a stall break at about 14 degrees, and a subsequent increase in lift. Whether this "final" lift increase is real or is an unsteady effect is unknown since the data was obtained via integration of the pressure coefficients, which were individually scanned and measured over a finite period of time. Nevertheless, the agreement in maximum lift coefficient is relatively good. At the lower angles-of-attack, the theory predicts slightly higher lift coefficients than measured in the tunnel. This slight discrepancy probably indicates that some small trailing edge separation existed in the actual experimental flow at angles-of-attack above six degrees which was not detected in the theory.

As indicated and discussed previously, the actual expermental lifts were obtained via pressure integration and resolved axial and normal force coefficients. Thus, perhaps a better comparison between theory and experiment would be to compare the theoretical lifts obtained via pressure integration with the wind tunnel values. Figure 6 compares the theoretical pressure lift coefficients (CLP) with the untripped LTPT data for these conditions. Again the agreement is reasonable with both sets of data predicting maximum lift coefficients of about 1.40. However, the CLP theoretical results indicate that miximum lift occurs at about 12 to 14 degrees angle-of-attack, while the stall break is hard to detect in this untripped experimental data.

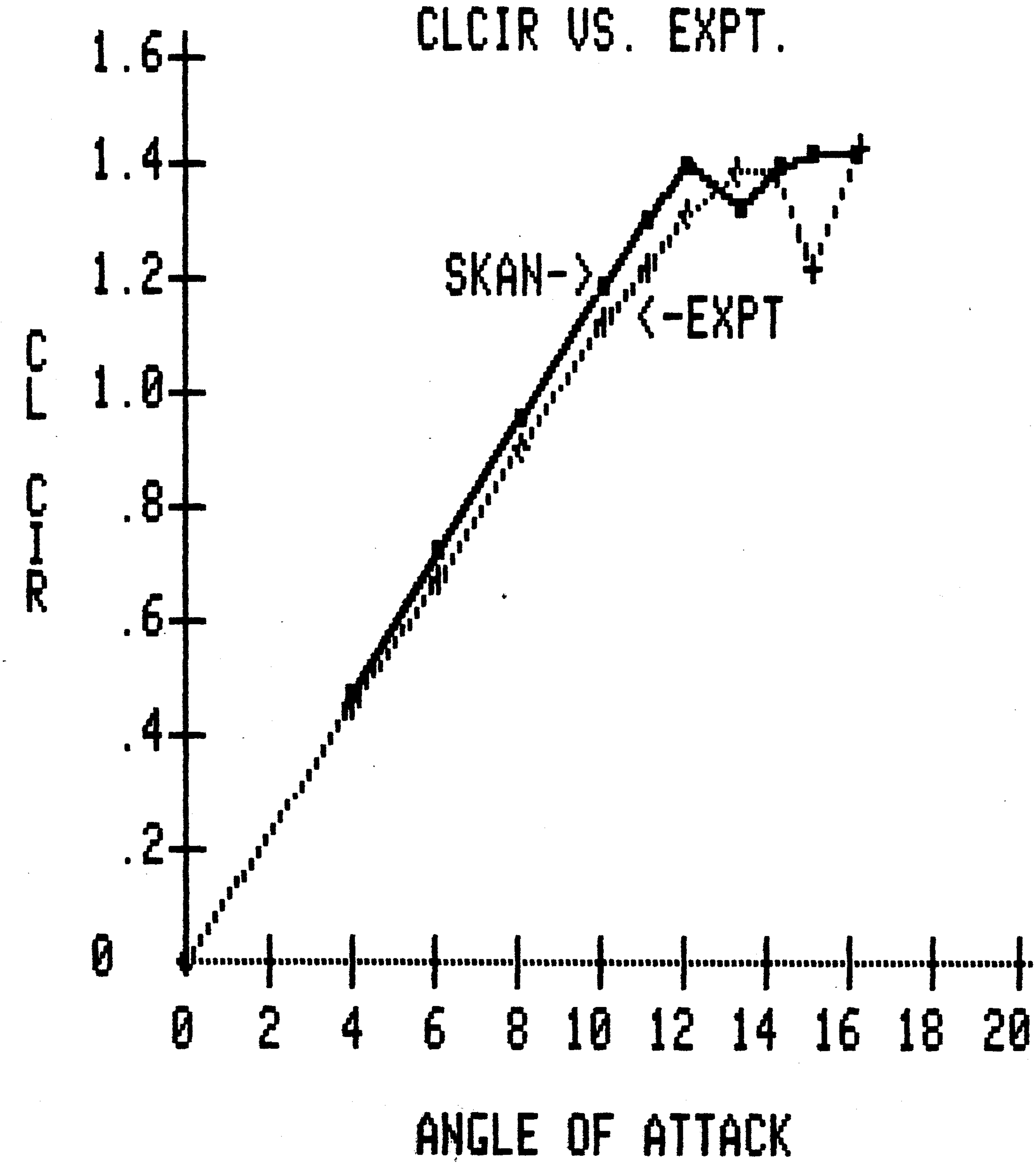

Figure 5 -- Comparison of SKAN Lifts with LTPT Data (NACA 0012, Mach No. = 0.3, Reynolds No. = 6 million)

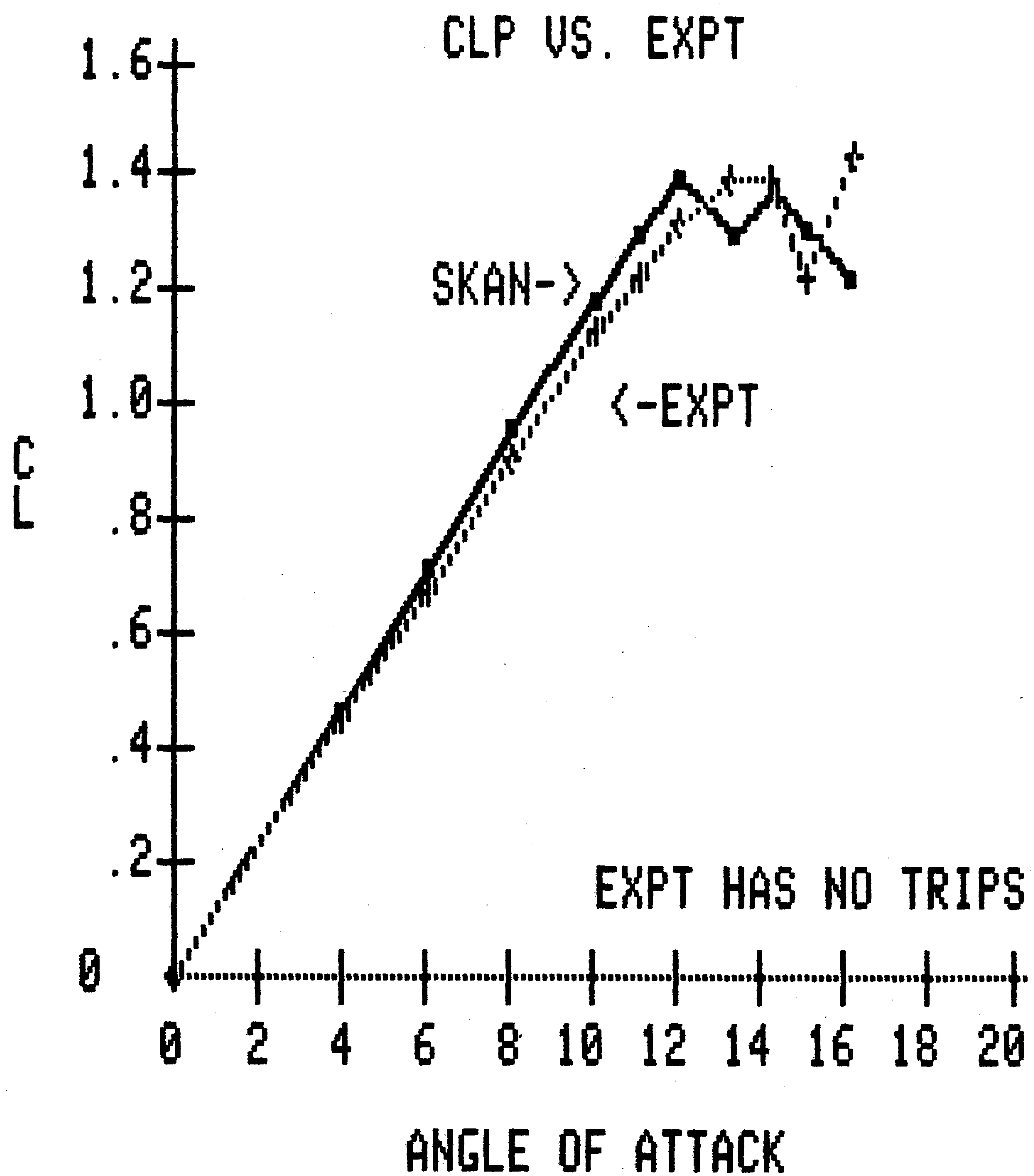

Figure 6 -- Comparison of SKAN CLP with Untripped LTPT Data
(NACA 0012, Mach No. = 0.3, Reynolds No. = 6 million)

In order to see the effect of trips on the experimental data, Figure 7 plots the theoretical CLP against the experimental data for the case in which boundary layer trips were used near the leading edge in the experiment. As can be seen on the figure, the agreement in values and trend is much better in this case. Both sets of data indicate a maximum lift coefficient of about 1.40 at about 14 degrees. The decrease in theoretical lift at about 13 degrees followed by a slight increase is probably due to the inherent inaccuracy in the location of the theoretical separation point. Also, notice that in this case, the experimental data indicates a definite stall break. Apparently, the usage of boundary layer trips in the experiment enhances the stability of the separation point near the leading edge in the high angle-of-attack cases. In spite of the slight discepancies between the theoretical and experimental results, it is believed that this figure demonstrates that the present theoretical method is a good approximation to massively separated flows at high angles-of-attack at medium Mach numbers.

At this point it should be noted that the present theoretical results are obtained using a laminar-turbulent boundary layer scheme with natural transition. However, at the high angles-of-attack of interest here, the theory consistently predicts upper surface transition at about one percent chord, which is in agreement with the forced tripping used in the wind tunnel experiments. However, the lower surface, because much of it experiences a favorable pressure gradient, frequently is theoretically laminar over most of its run. Since it is believed that trips were also used in the experiment on the lower surfaces, the lower surface pressures might be slightly different. However, numerous numerical tests indicate that the difference in predicted lift coefficients between early forced transition and natural transition on the lower surface is

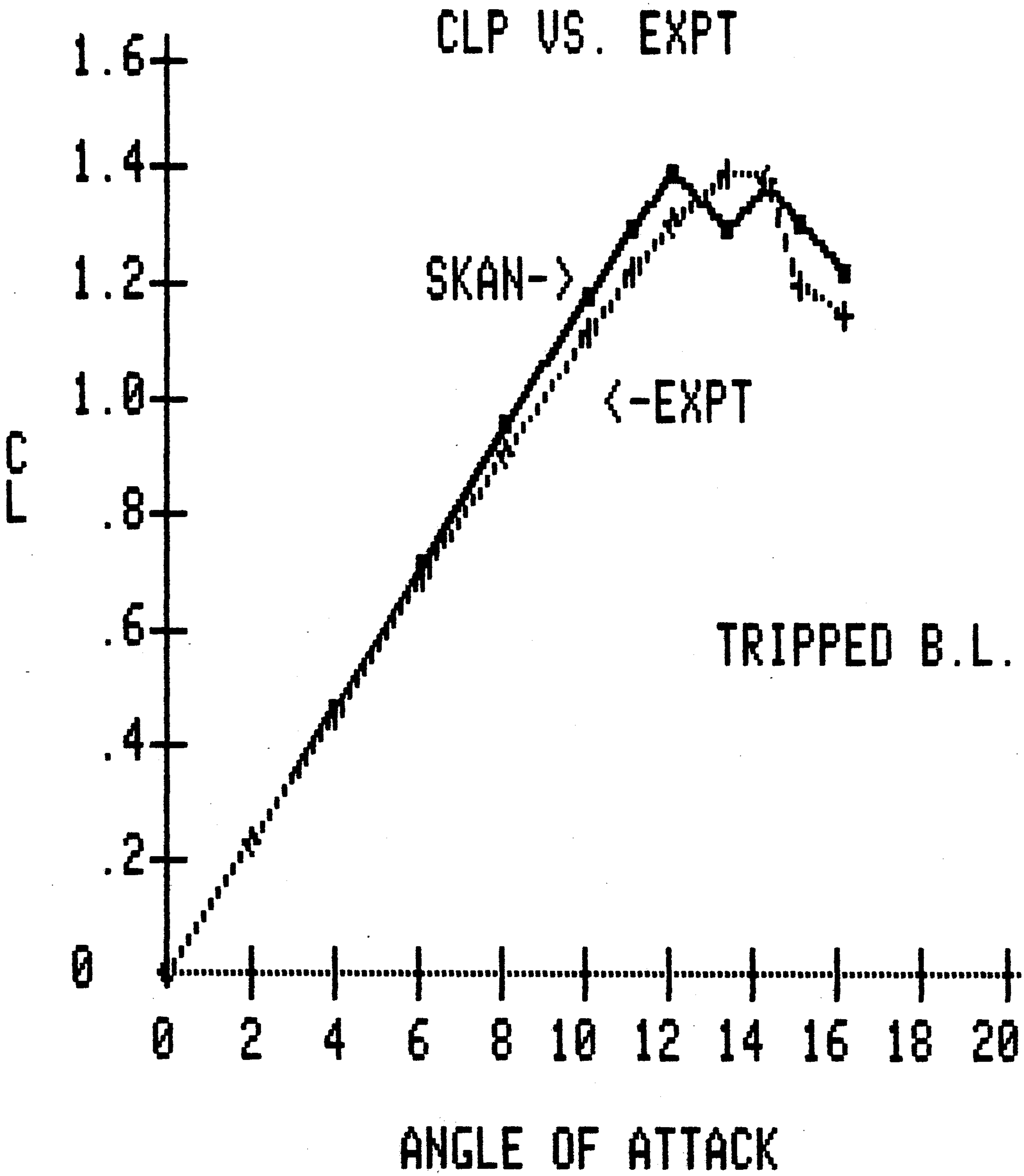

Figure 7 -- Comparison of SKAN CLP with Tripped LTPT Data
(NACA 0012, Mach No. = 0.3, Reynolds No. = 6 million)

negligible. Thus, the present theoretical results should be a reasonable representation of the tripped boundary layer experiments. The only effect of the lower surface trips should be on the values of the drag coefficients, where it would be expected that the tripped cases would have higher drags than the untripped cases.

Similarly, Figure 8 also shows the experimental data for the case in which the boundary layer is tripped; but here it is compared to the theoretical results obtained from SKAN using the constant separated pressure option (i.e. KSEP=0). As can be seen, the agreement is not as good as in the previous figures. While both the experimental data and the KSEP=0 theory indicate about the same maximum lift coefficient, the theoretical lift curve peaks at about 12 degrees instead of around thirteen to fourteen degrees. Also, the theoretical lift decreases very rapidly in the post stall region. It should be noted that no theoretical value is plotted for sixteen degrees angle-of-attack since the KSEP=0 option blew up for this case due to the movement of the separation point all the way to the leading edge. At the point of numerical failure, which occurred about half way thru the fine grid (97x49), the lift coefficient was down to 0.8 and was decreasing very rapidly.

In developing a model for high angle-of-attack massively separated flow, it would be desirable not only to be able to predict the lift but also to be able to predict the drag. Figure 9 presents drag polars for the present theoretical data (NACA 0012, Mach = 0.3, and Reynolds Number = 6 million) and for data obtained in the LTPT tunnel with boundary layer trips on the airfoil model. In the pre-stall region, the experimental drag is slightly higher than the theoretical values; and the fact that it starts to increase earlier (i.e. at CL of 0.9) than the theoretically predicted drag indictes that some trailing edge

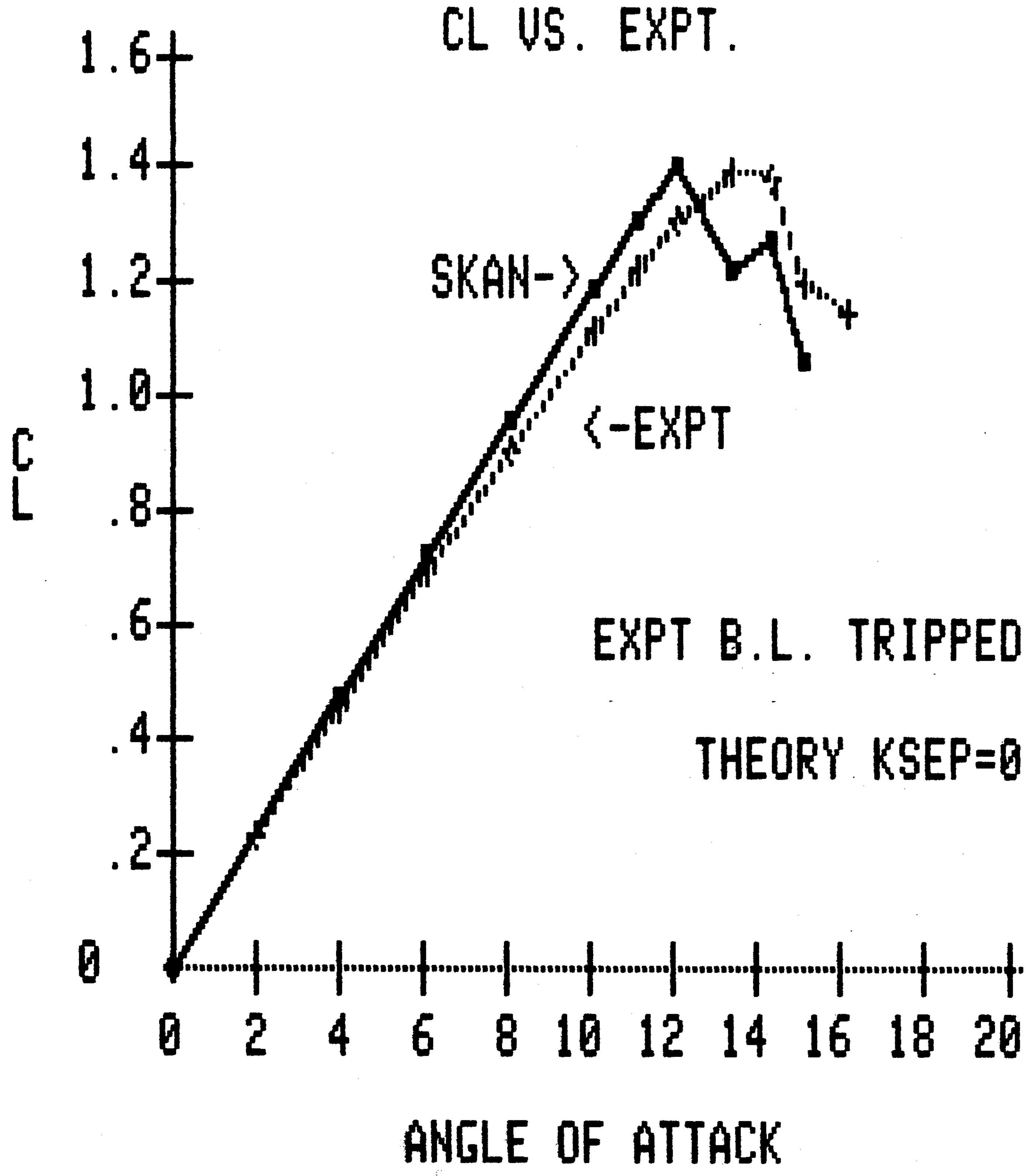

Figure 8 -- Comparison of SKAN KSEP Zero Results with LTPT Data
(NACA 0012, Mach No. = 0.3, Reynolds No. = 6 million)

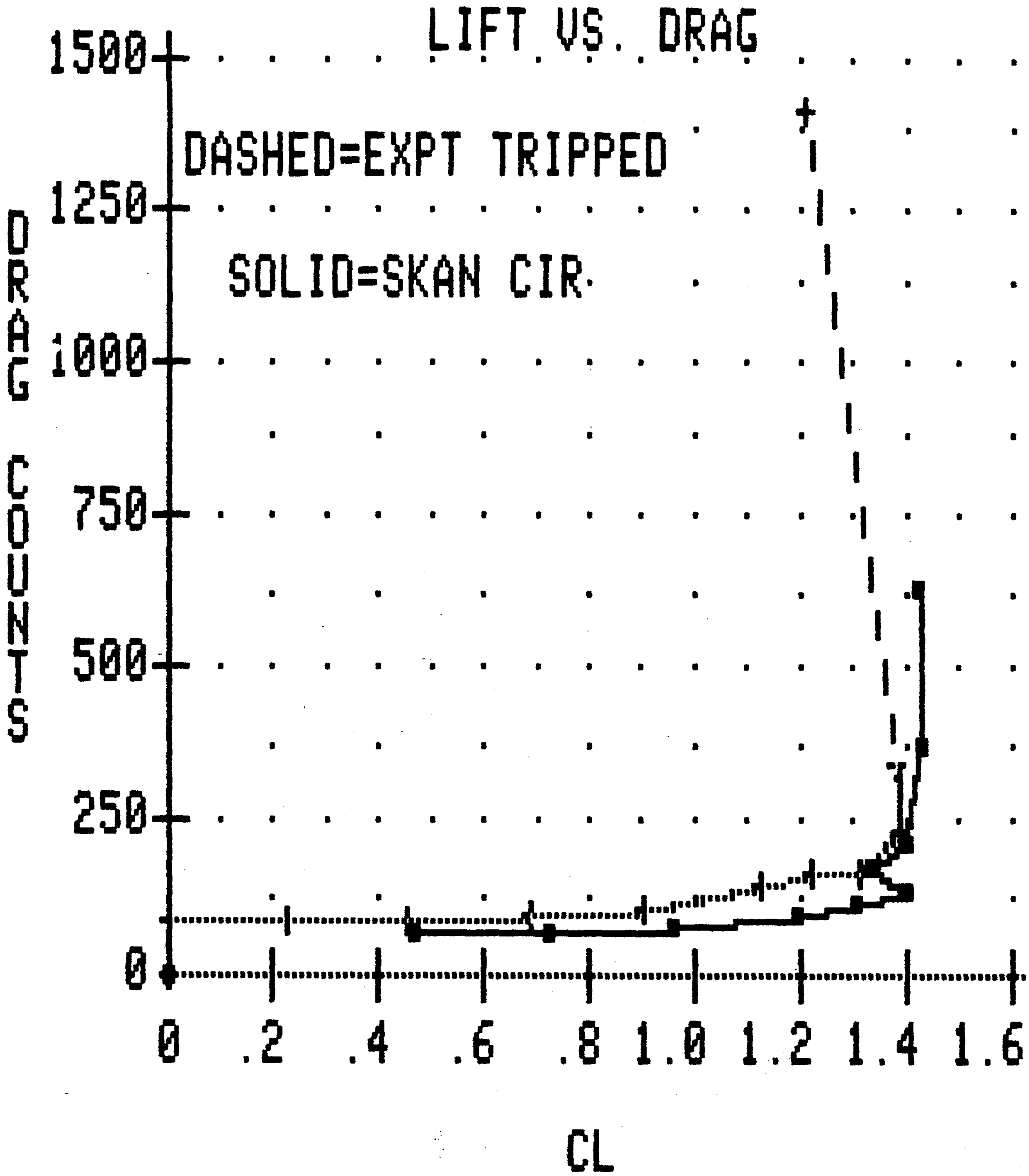

Figure 9 -- Drag Polars for SKANSEP and Tripped LTPT Data
(NACA 0012, Mach No. = 0.3, Reynolds No. = 6 million)

separation existed in the experiments where none was predicted by the theoretical results. Nevertheless, the location and shape of the drag break is reasonably well predicted by the theoretical results. However, after maximum lift, the experimental drag values are consistently larger than the theoretical results. On these and subsequent figures, it should be noted that the drag coefficient is plotted as drag counts, where 100 drag counts is equivalent to a drag coefficient of 0.0100.

As discussed previously, since the theoretical results assume natural transition, the theoretical results for drag should in general be lower than the experimentally tripped data and be in better agreement with the experimental values obtained using clean airfoils. Figure 10 shows that this situation is indeed the case. Here the theoretical drag polar is compared to LTPT data obtained without using boundary trips on the airfoil model. As can be seen, the agreement between the theory and the experiment at low values of lift coefficient is significantly better than on the previous figure and is excellent. Also the drag break is reasonably well predicted. It should be noted that on Figures 9 and 10, the theoretical lift coefficient value was that obtained using the circulation; while the theoretical drag values were those obtained from a modified Squire-Young formula.

For completeness, Figure 11 portrays the same experimental data and theoretical lift except data that the latter was plotted using the lift coefficient obtained from integration of the pressure distributions. Again the overall agreement is acceptable. It is believed that Figures 10 and 11 demonstrate that the present model of SKANSEP can be used to predict reasonably accurate values of drag for airfoils having massive separation at medium Mach numbers.

Up to now, results have been presented which demonstrate that the present theoretical model can yield good predictions for the aerodynamic coefficients.

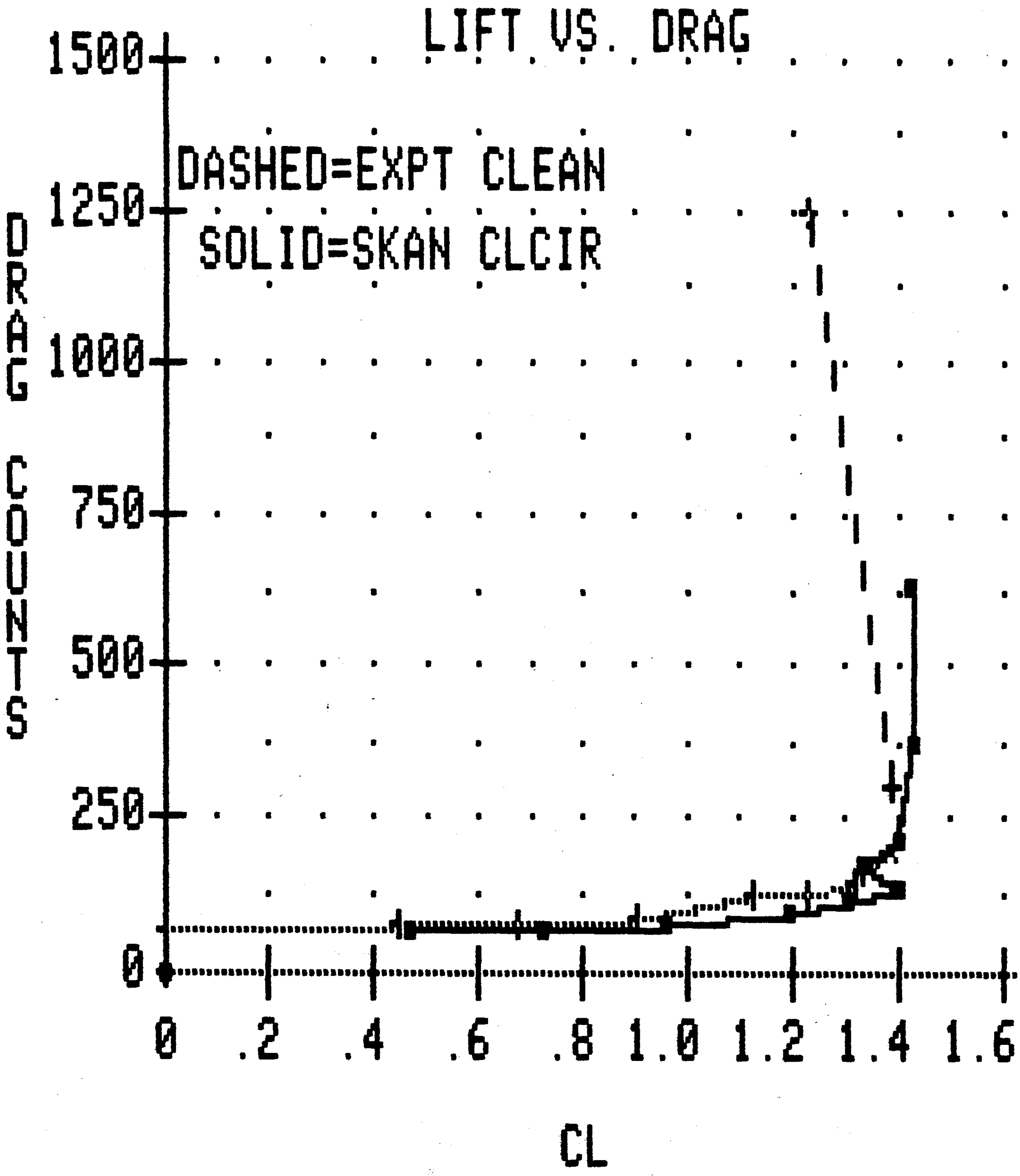

Figure 10 -- Drag Polars for SKANSEP and Untripped LTPT Data
(NACA 0012, Mach No. = 0.3, Reynolds No. = 6 million)

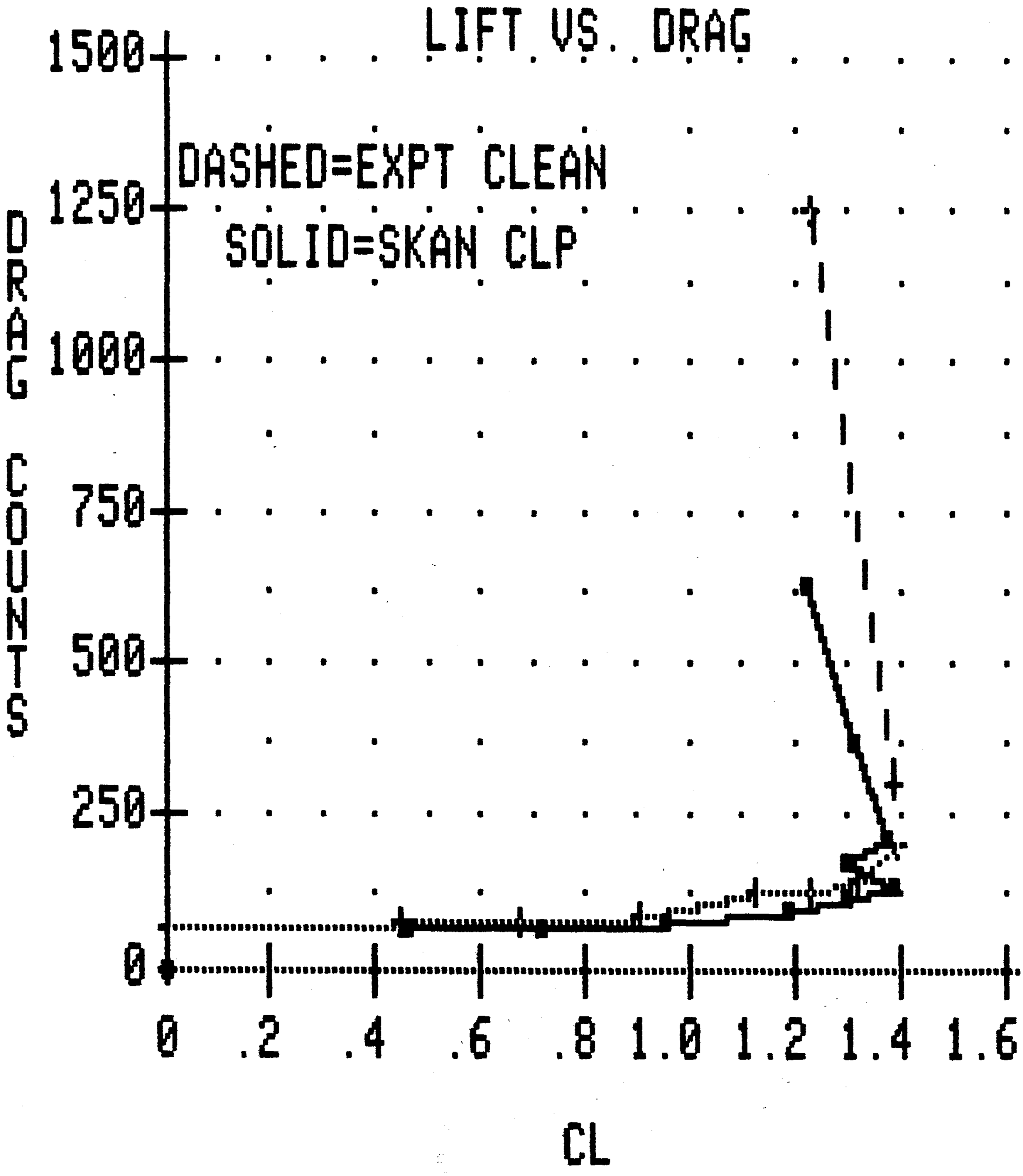

Figure 11 -- Drag Polars for SKAN using CLP and Untripped LTPT Data
(NACA 0012, Mach No. = 0.3, Reynolds No. = 6 million)

However, since these values are obtained via integration, it is possible to have such good agreement in overall coefficients while at the same time having serious disagreement between theory and experiment in the actual pressure distributions. Thus, the next series of figures present experimental pressure distributions from the LTPT tunnel and theoretical results obtained using SKANSEP with the variable separated pressure option (KSEP=1). Unfortunately, plotting both the experimental and theoretical results on the same figure would lead to confusing plots which are difficult to interpret. Thus, the experimental data and the theoretical data are plotted separately.

Figure 12 (a) presents experimentally measured results for a NACA 0012 at 12.09 degrees angle-of-attack. These results were obtained in the Langley LTPT wind tunnel, had twenty-five pressure ports on each surface of the airfoil, and should be compared to the theoretical distribution presented in Figure 12 (b). The latter was obtained using the corrected Mach number, angle-of-attack, and Reynolds number determined for the experimental case. As can be seen by comparing the two figures, both predict about the same minimum pressure coefficient on the upper surface; and both have, within plotting accuracy, the same values and trend in the trailing edge region. For this case, the theoretical result predicted upper surface transition with a short separation bubble at one-percent chord with no trailing edge separation. Since natural transition was assumed in the theoretical model, lower surface instability was predicted at 59% chord followed by transition to turbulent flow at 95% chord. It should be noted that the theoretical model did not predict any upper surface separation prior to 99 percent chord. The latter is the last point computed on the airfoil surface.

Figures 13 (a) and 13 (b) present similar results at 13.4 degrees angle-of-attack. Here again the overall agreement is in general very good;

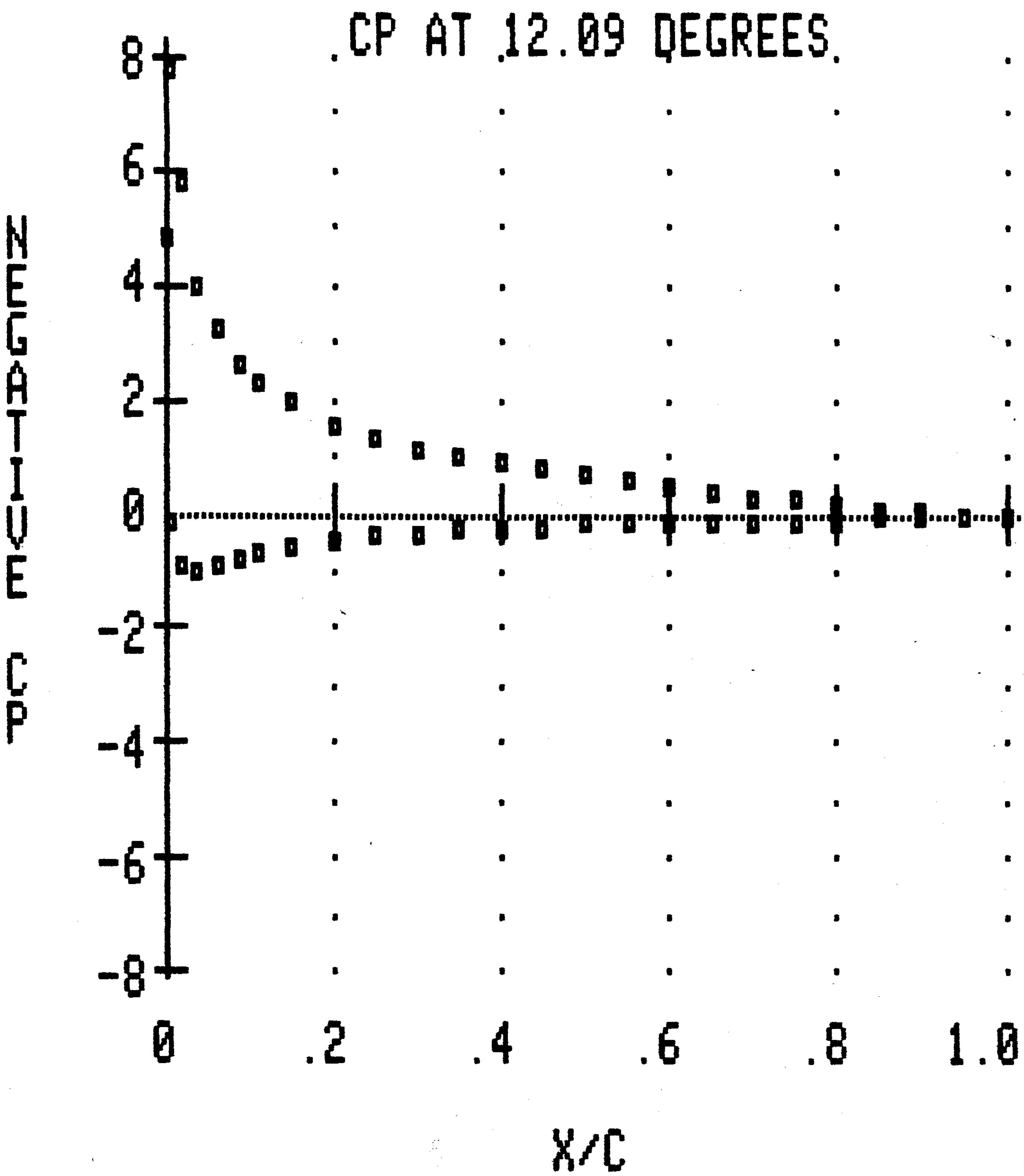

Figure 12(a) -- Pressure Distribution for NACA 012 from LTPT Data (Mach no. 0.302, AOA 12.09 deg., Reynolds No. 5.96 million)

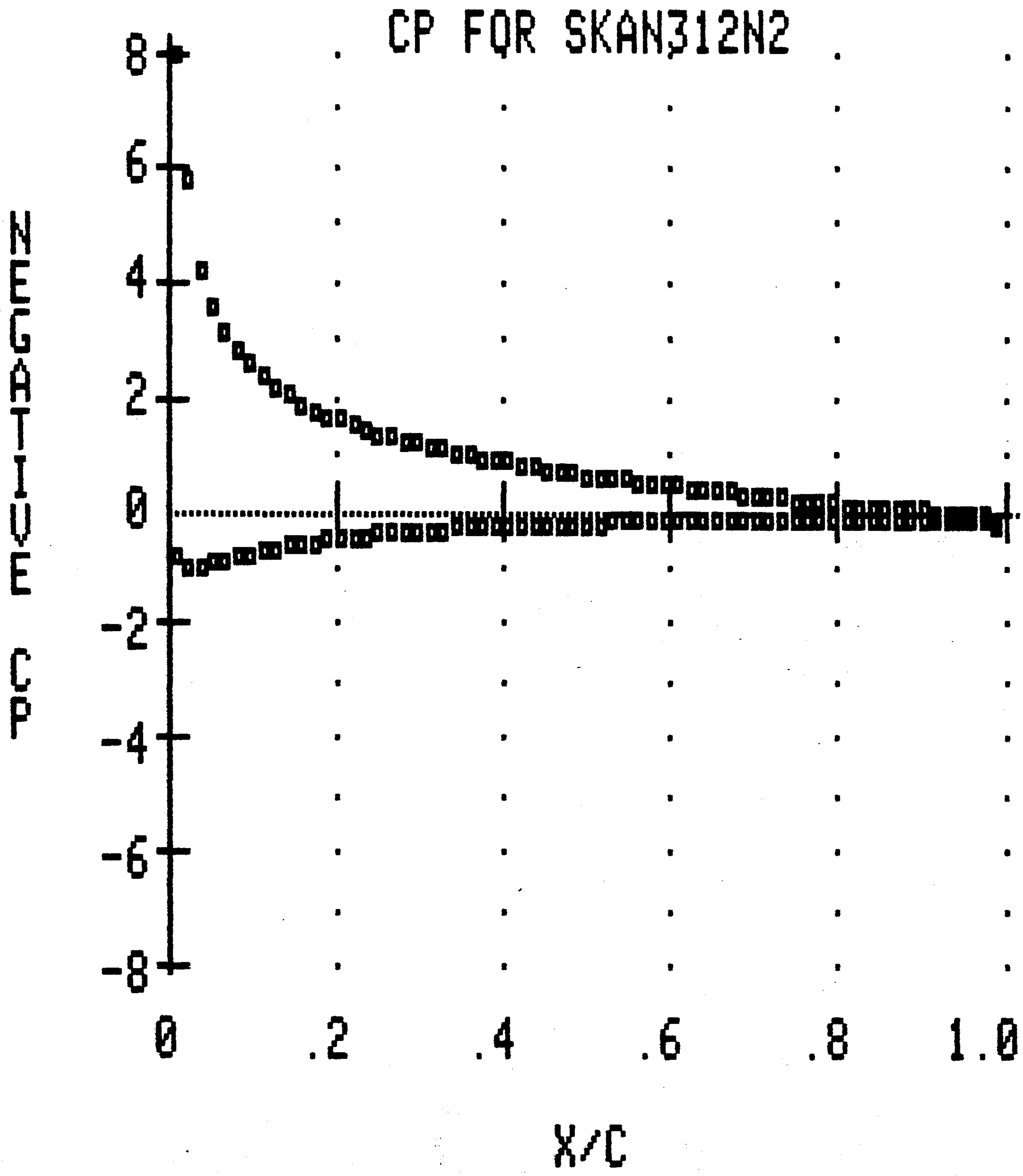

Figure 12(b) -- Theoretical CP Distribution for NACA 0012
(Mach No. 0.302, AOA 12.09 deg., Reynolds No. 5.96 million)

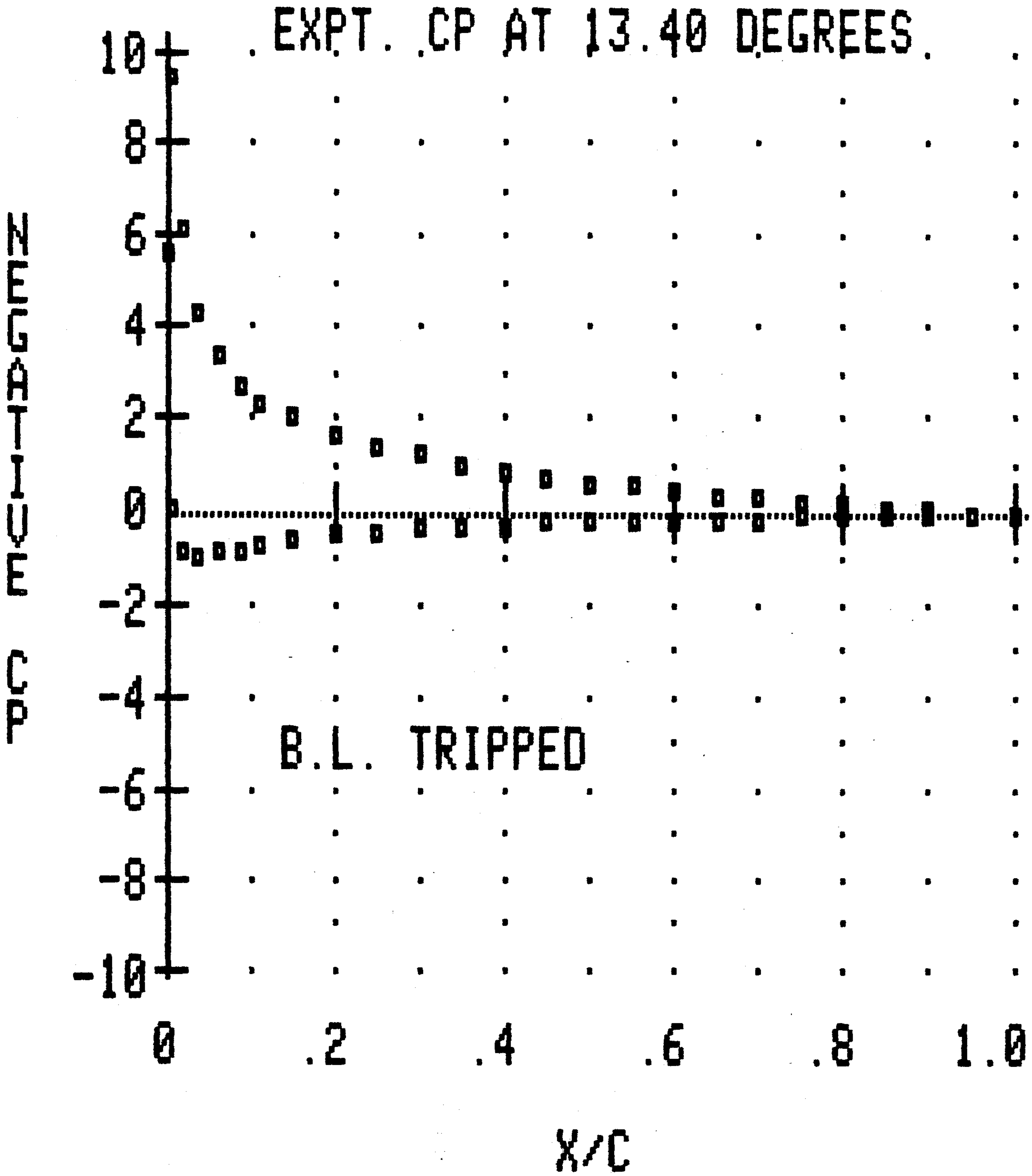

Figure 13(a) -- Pressure Distribution for NACA 0012 from LTPT Data
(Mach No. 0.301, AOA 13.4 deg., Reynolds No. 5.98 million)

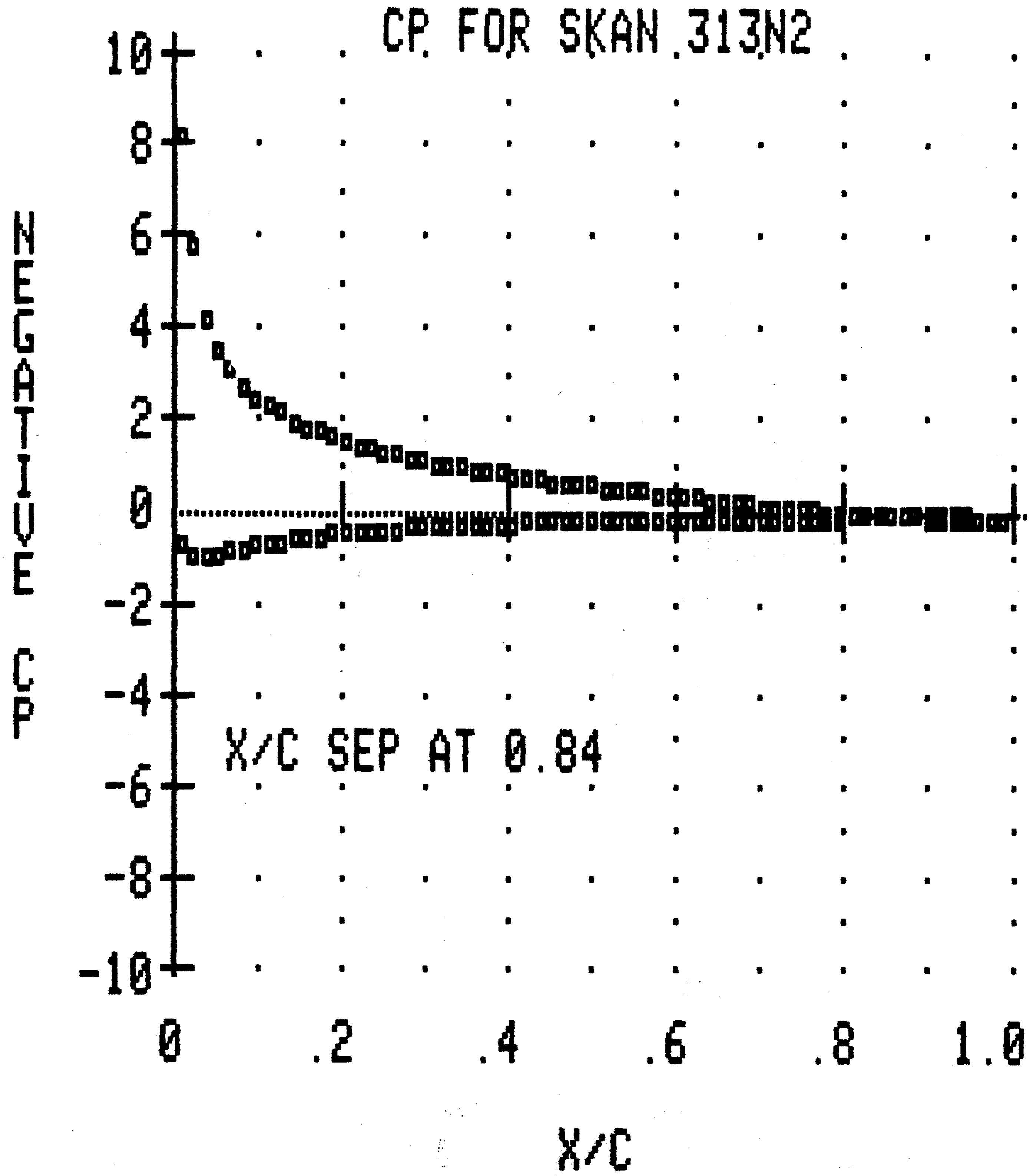

Figure 13(b) -- Theoretical CP Distribution for NACA 0012
(Mach No. 0.301, AOA 13.4 deg., Reynolds No. 5.98 million)

although there appears to be some discrepancy in the value of minimum pressure coefficient measured experimentally and predicted theoretically. However, superposition of the two plots indicates that the agreement is excellent. The apparent discrepancy is simply due to the fact that the experiment measured points closer to the leading edge than is possible with the grid used in the numerical calculation. For this case, the theory predicted lower surface instability at 64 percent chord followed by turbulent transition at 96 percent chord. On the upper surface, transition to turbulent flow via a short separation bubble occurred at one percent chord and turbulent separation occurred at 84 percent chord.

Figures 14 (a) and 14(b) compare experimental results obtained at 14.31 degrees angle-of-attack, which corresponds to the conditions of maximum lift in the experiment, with theoretical values obtained using SKANSEP with the variable separated pressure option (KSEP=1). Here, in the vicinity of the trailing edge, the experimental data for the upper surface indicates slightly negative values for the pressure coefficient, while the theoretical results are always slightly higher. Nevertheless, the overall agreement is excellent. For this case, the theory predicted lower surface instability at 76 percent and turbulent transition at 97.5 percent chord. On the upper surface, transition occurred at one percent chord with separation being at 81 percent. It should be noted that the experimental data presented here is for the case in which boundary layer trips were used on the airfoil model. However, as mentioned previously, the difference in pressure coefficients between the tripped and untripped cases is negligible.

Figure 14 (c) also shows theoretical results obtained at 14.31 degrees angle-of-attack, but here the constant separation pressure option (KSEP=0) was used. As can be seen by comparing Figures 14 (a) and 14 (c), the latter

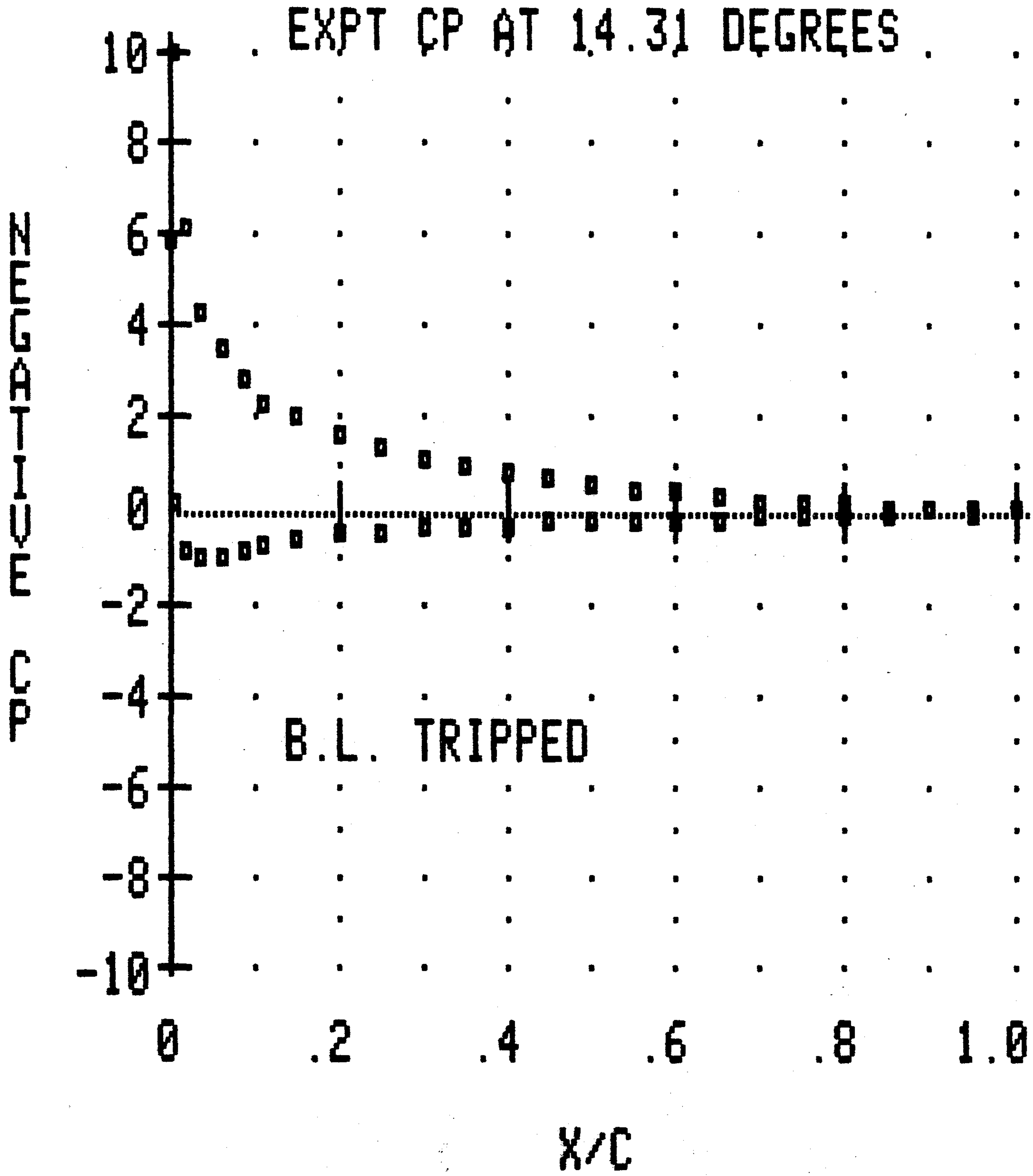

Figure 14(a) -- Pressure Distribution for NACA 0012 from LTPT Data (Mach No. 0.301, AOA 14.31 deg., Reynolds No. 5.97 million)

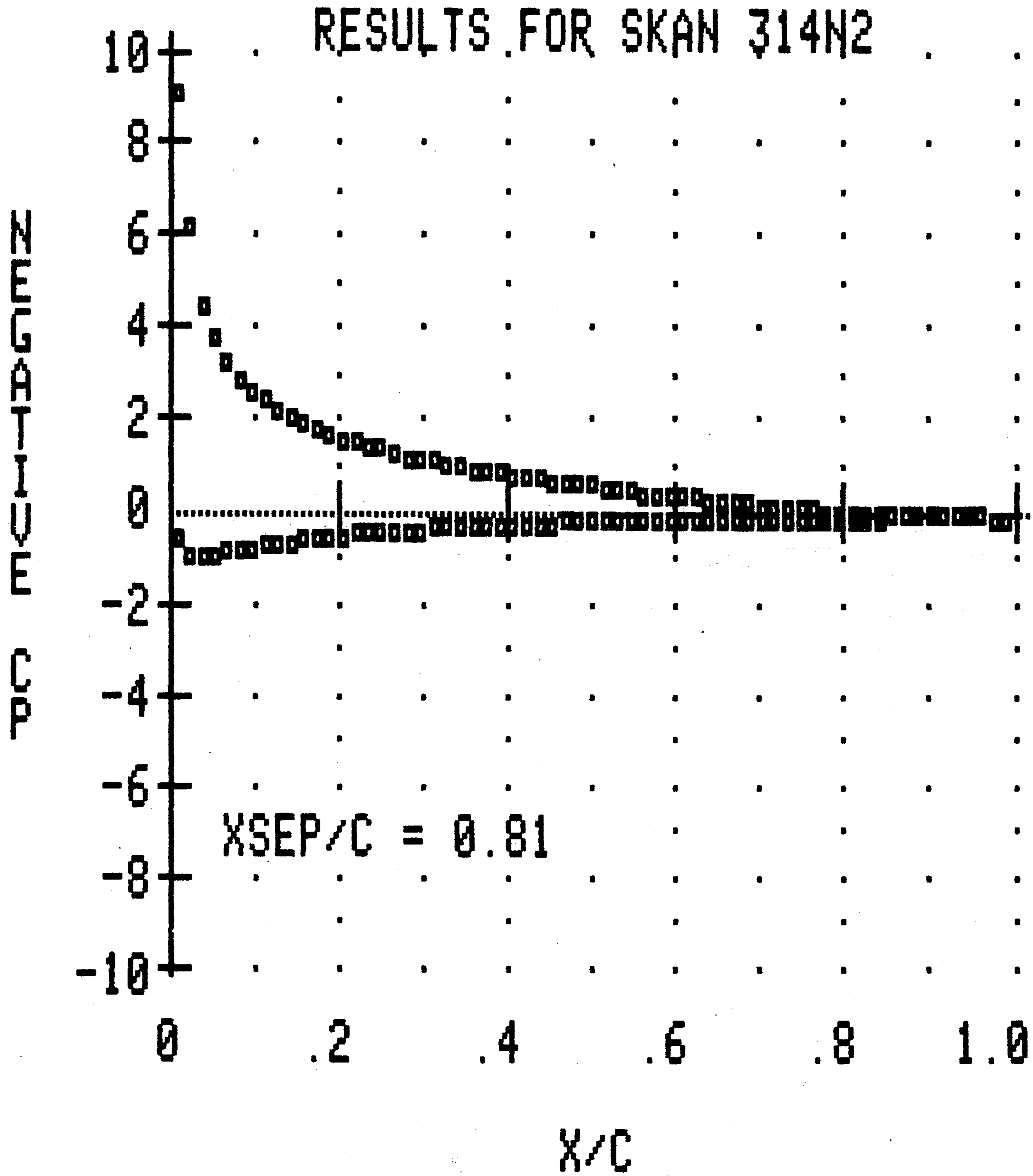

Figure 14(b) -- Theoretical CP Distribution for NACA 0012
(Mach No. 0.301, AOA 14.31 deg., Reynolds No. 5.97 million)

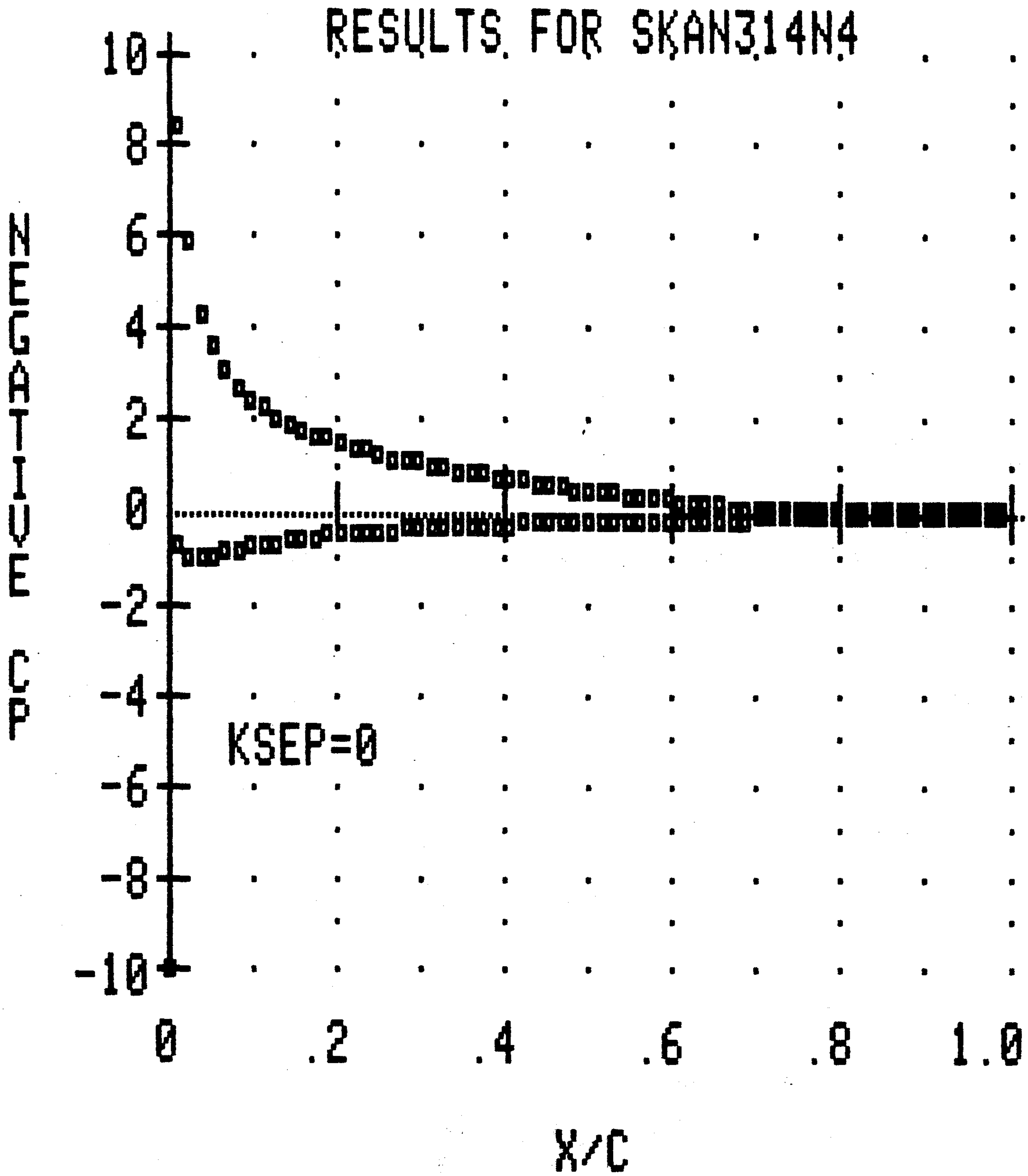

Figure 14(c) -- Theoretical CP Distribution for NACA 0012 Using KSEP = 0 Option
(Mach No. 0.301, AOA 14.31 deg., Reynolds No. 5.97 million)

"underpredicts" the minimum pressure coefficient, but it does yield pressures in the vicinity of the trailing edge which are in better agreement with the experimental data. However, the predicted lift coefficient for this case is considerably lower than the experimental value. Thus, in spite of the apparent better agreement near the trailing edge, it is believed that the KSEP=1 model is still the better model for these conditions.

For this case, the run corresponding to Figure 14 (c) predicted for the lower surface laminar instability at 92 percent chord with no transition to turbulent flow before the trailing edge. For the upper surface, it predicted short bubble transition at one percent chord followed by separation at 75 percent chord.

The final set of figures, Figures 15 (a) and 15 (b), compare experimental and theoretical (KSEP=1) results at a nominal angle of attack of 15 degrees. This angle is above the maximum lift coefficient and is into what normally would be termed the stall regime. For this case, the agreement between experiment and theory is not all that good. However, considering the unsteady nature of the actual flow, indicated by the oscillations in the experimental data, and the unknown effects of wind-tunnel walls at these conditions, the agreement is probably acceptable, particularly considering the good agreement in lift coefficients shown on Figure 7. For this case, the theory indicated lower surface laminar instability at 76 percent chord with full transition at 97 percent. On the upper surface, transition occurred at one percent chord followed by turbulent separation at 51.6 percent. Thus, for this case, the flow over approximately half of the airfoil on the upper surface was separated. If this extensive and massive separation is considered when comparing the theoretical values with the experimental data, the general agreement of the results are probably acceptable, if not surprising.

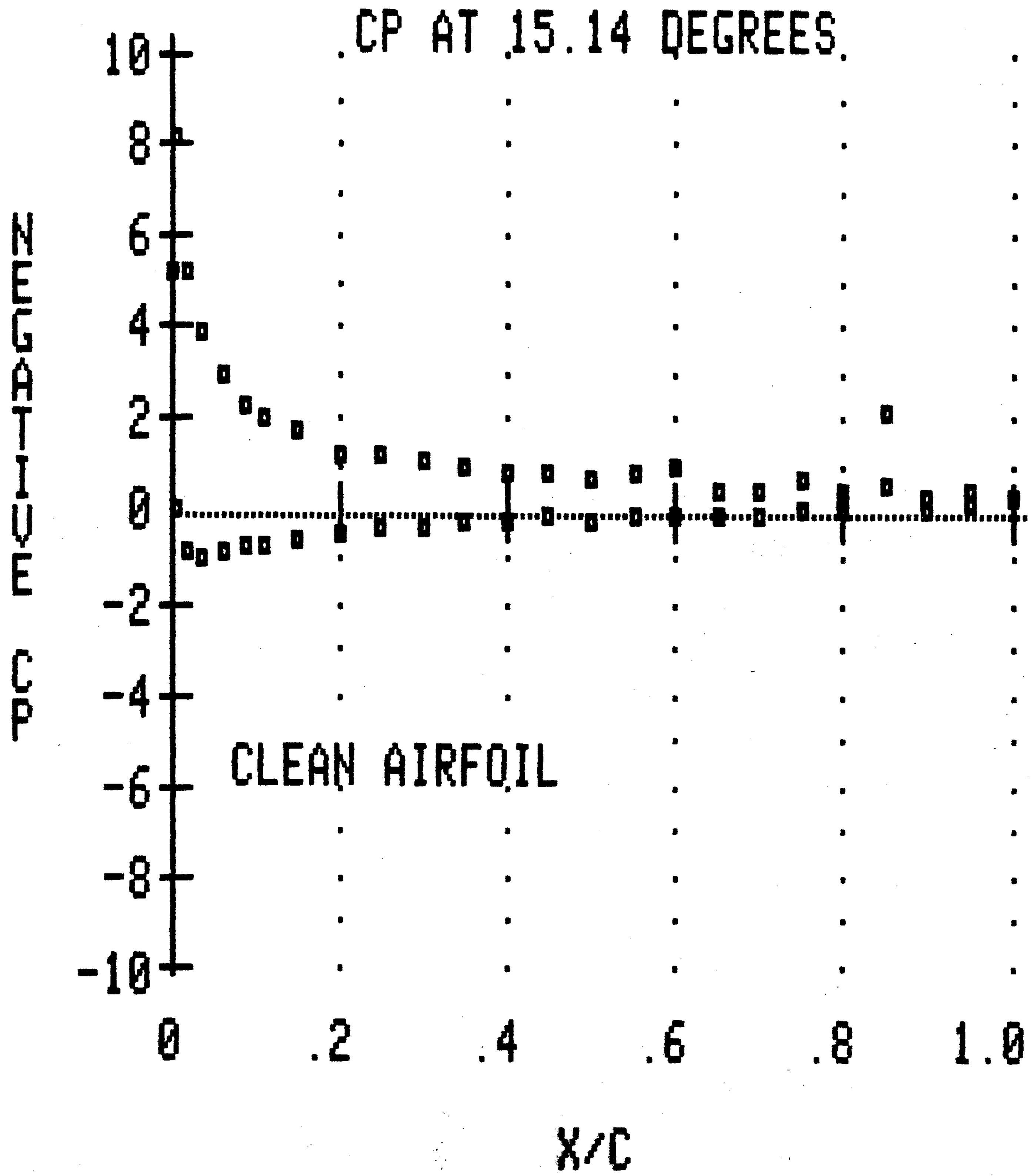

Figure 15(a) -- Pressure Distribution for NACA 0012 from LTPT Untripped LTPT Data (Mach No. 0.301, AOA 15.14 deg., Reynolds No. 5.94 million)

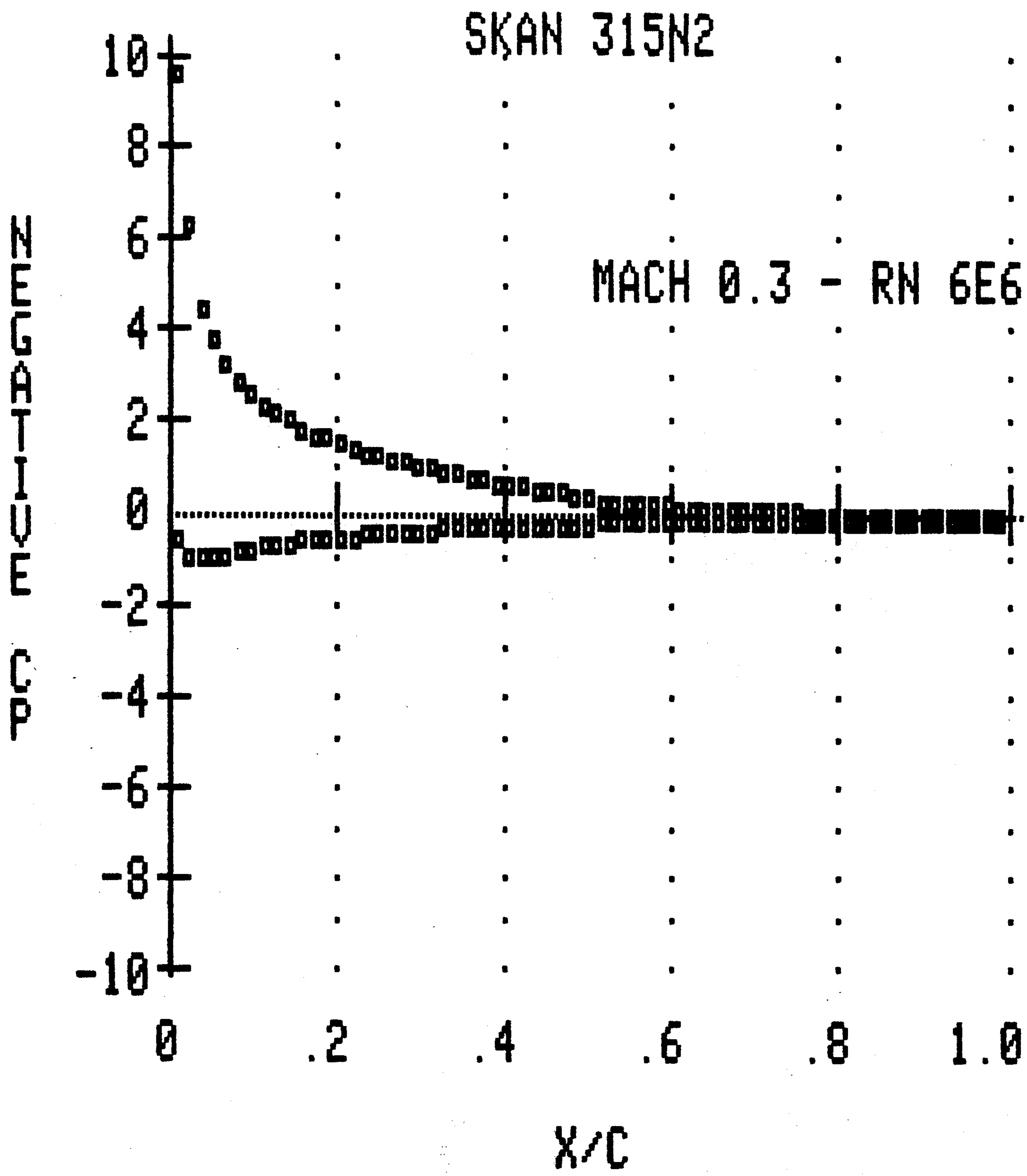

Figure 15(b) -- Theoretical CP Distribution for NACA 0012 Airfoil
(Mach No. 0.304, AOA 15.08 deg., Reynolds No. 6.05 million)

It is believed that the results presented in Figures 12 thru 15 demonstrate that at least at medium freestream Mach numbers the present version of SKANSEP using the KSEP=1 option can yield accurate predictions of list coefficient behavior, drag variation with lift, and airfoil surface pressure distributions.

IV. Summary

During the present reporting period, the SKANSEP computer code was used to investigate massive separated flow at medium Mach numbers. Based on the results presented in this report, it is believed that SKANSEP can be used to predict reasonably accurate results for high angle-of-attack flows about airfoils at medium subcritical Mach numbers. During the next reporting period, this new version of SKANSEP will be used to investigate flows at supercritical Mach numbers and, hopefully, the code will be validated for those conditions.

V. References

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2. Barnwell, R.W., "A Potential-Flow/Boundary-Layer Method for Calculating Subsonic and Transonic Airfoil Flow with Trailing Edge Separation". NASA TM-81850, June 1981.
3. Anderson, W.K. Thomas, J.L. and Rumsey, C.L., "Application of Thin Layer Navier-Stokes Equations Near Maximum Lift," AIAA Paper No. 84-0049, 22nd Aerospace Sciences Meeting, January 1984.